Landsat-6 Failure Investigation Final Report Summary

January 16, 1995

National Oceanic and Atmospheric Administration (NOAA)
Landsat-6 Failure Review Board

Thomas E. McGunigal, Chairman

Martin Marietta Corporation
Investigation Board

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1.0 INTRODUCTION

On October 5, 1993, at 17:56:29 UTC, the Landsat-6 spacecraft was launched from Vandenberg Air Force Base (VAFB) on the G5 Titan II booster. Throughout the powered flight, all telemetered data from the Titan II was nominal. The data indicated that a spacecraft separation from the booster occurred at the nominal time and place. All expectations were that Landsat-6 would be contacted at the first ground station, Kiruna, Sweden, at approximately 19:06 UTC. The spacecraft never contacted the station. Attempts to locate the satellite were futile, and other assets reported reentry events at locations downrange from the observed booster reentry events. It was concluded that Landsat-6 had not achieved orbit after separation from the Titan II.

Immediately after the Landsat-6 failure, the Martin Marietta Corporation began assembling senior technical and executive staff members to form its investigative board. Similarly, the National Oceanic and Atmospheric Administration began assembling its review board. The Martin Marietta Board was chaired by Robert J. Polutchko and included representatives from the National Oceanic and Atmospheric Administration (NOAA), the National Aeronautics and Space Administration (NASA), the Earth Observation Satellite Company (EOSAT), and a broad cross section of Martin Marietta senior personnel. The first Martin Marietta Board meeting was held on October 11, 1993. The board adopted the following goals for the investigation:

- Identify the root cause, or most probable cause of the loss of Landsat-6;
- Identify corrective actions needed to preclude a reoccurrence of the identified cause on other spacecraft products;
- Identify any and all deficiencies associated with the production of Landsat-6 discovered by the investigative process; and,
- Recommend corrective actions for those deficiencies.

On October 22, members of the Martin Marietta Investigative Board and members of the Martin Marietta Landsat-6 Program Office met with members of the NOAA Investigative Board. The NOAA Board was chaired by Thomas E. McGunigal and included representatives from the U.S. Air Force DMSP and Landsat-7 Programs, the Aerospace Corporation, MIT Lincoln Laboratory, NASA, EOSAT and several NOAA members. The membership of both investigative boards is shown in Section 10.0. At this meeting, Martin Marietta briefed the NOAA Board on the information accumulated to date and on the investigative process already underway at Martin Marietta. The Martin Marietta Board invited the NOAA Board to share all information and conduct cooperative meetings with the Martin Marietta Landsat-6 Project office while maintaining board independence. The initial Joint NOAA-Martin Marietta Investigative Board meeting was held on October 28 and 29, 1993.

The Boards reserved the right to conduct independent investigation activities and to develop independent findings and reports, as appropriate. However, as the investigation evolved, the effective participation of both Boards in the work of the investigation team led to a full consensus on the findings summarized in this report.

Therefore, this report documents the summary findings of an eight-month investigation into the causes of the Landsat-6 failure by the joint investigation efforts of the NOAA-sponsored Government Investigative Board and a Martin Marietta Corporation Investigative Board. The conclusions of this report were made possible by the close cooperation of all the participants, who represented a broad range of aerospace experience and expertise. Major contributors include investigators from the Aerospace Corporation, the Martin Marietta Engineering Propulsion Labs (Denver), the Martin Marietta Government Electronic Systems Division, McDonnell Douglas Aerospace-Space Systems, Olin Aerospace/Rocket Research Corporation, Thiokol Corporation, and the U.S. Air Force. Special recognition should be given to the White Sands Missile Range for their contributions to the investigative process and the results published in this report. The results and conclusions of this report also incorporate contributions from the Astro Space Landsat-6 Program Office as documented in the Landsat-6 Program Office final reports, voluminous meeting documentation and analyses.

This investigation concludes that the Landsat-6 spacecraft experienced a ruptured hydrazine manifold of the Reaction Control System (RCS). The ruptured hydrazine manifold rendered the spacecraft's reaction engine assemblies (REAs) useless because fuel could not reach the engines. As a consequence, there was a failure to maintain attitude control during the apogee kick motor (AKM) burn. This failure caused the spacecraft to tumble during the AKM burn and not accumulate sufficient energy to attain orbit. The spacecraft reentered the atmosphere south of the equator 1808 seconds after liftoff. The reentry of the satellite is validated by the lack of a Landsat signal at Kiruna, Sweden and by the observations of other national assets.

The ruptured hydrazine system was first detected as a large shock signature as sensed by the booster instrumentation package, secondly as the cause of a low separation velocity as determined by the same sensor package and data from the MOTR radar and finally, as a loss of attitude control as initially determined by the MOTR radar and then by the lack of signal at Kiruna and by the observations of other national assets. The effects of a ruptured manifold satisfied all the observations and met the criteria of the investigators and Board members.

The most probable cause of that rupture was an unpredictable hydrazine detonation mechanism that occurred during the activation of a normally closed pyrovalve. This detonation mechanism occurred in one of two post flight hydrazine tests of a complete hyrdazine system mockup. The term "unpredictable mechanism" calls attention to the fact that the exact cause of the detonation has not been determined. A failure of the 1/2 inch hydrazine lines at the location of the valve has been shown, by analysis, to be consistent with the observed effects. It is reasonable to conclude that the Landsat-6 failure was due to an explosive event in the hydrazine system caused by conditions not previously reported as to be capable of triggering detonation of hydrazine.

The findings of the Joint Investigative Board must be applied to future projects with maximum emphasis on the lessons learned from this loss. Suggestions are provided in Section 8.0 and 9.0, Conclusions and Recommendations respectively, for improvements in testing and modeling hydrazine fuel systems.

2.0 THE SPACECRAFT DESIGN

As with the earlier Landsat spacecraft, the mission design called for a polar orbit at approximately 705 kilometers altitude. In the case of Landsat-6, this meant that the spacecraft would be launched on a Titan II from VAFB. This site is used to launch high inclination orbiters since it allows launching in a southerly direction, onto a long downrange that is largely uninhabited, and therefore, presents low risk to human life in the event of a launch mishap. As a result of payload weight-to-orbit versus booster cost trade-offs, it was decided that the Landsat-6 spacecraft would utilize an insertion strategy similar to the DMSP and TIROS spacecraft. Therefore, the spacecraft was designed to circularize its own orbit using a Thiokol Star 37XFP solid rocket AKM, and was outfitted with a propulsion system (alternately called the Reaction Control System, or RCS) similar to that used on the DMSP and TIROS series spacecraft. However, modifications to the heritage spacecraft RCS were required. The heritage spacecraft RCS needed only to control the ascent phase, and provide clean propulsive forces (GN_2 thrusters) for quelling any large unexpected attitude disturbances on orbit. However, the Landsat-6 ground-track repeat requirements necessitated a system capability for propulsive efforts throughout the mission. The resultant Landsat-6 RCS configuration was an integrated system that comprised elements of the heritage systems and smaller monopropellant hydrazine engines, called Orbit Adjust Engines (OAEs).

From the standpoint of both spacecraft and mission design, the mission was divided into three major phases: the Ascent Phase, the Early-Orbit Phase, and the Operational Phase. Since the spacecraft failed to achieve orbit, only the ascent phase is discussed in this summary report.

The Ascent Phase of the mission is depicted in Figure 1. The ascent from the launch pad through nominal orbit insertion, and culminating in solar array deployment, proper attitude attainment, and handover to the Orbit Mode Software (OMS) was planned to have taken 40 minutes. During this phase the spacecraft operated under the control of the Ascent Guidance Software (AGS). This software used the Inertial Measurement Unit (IMU) to navigate during the booster guided portion of the flight. After effecting separation from the booster, the AGS was to guide the spacecraft into the nominal orbit, or the best attainable orbit, based on the availability of useful input data. After firing the AKM and providing the final velocity trim, the REAs were to be closed off and the Solar Array deployed. The Ascent Phase included actuating several pyrotechnic devices: the two normally closed system pyro valves (PV-1 and PV-2) as shown in Figure 5, the four clamp band retaining nuts, the REA isolation valve (IVH), and the 12 cable cutters used to sever the six Solar Array retaining cables. The Ascent Phase was conducted with the spacecraft RF silent, since the S-band telemetry transmitter operated at the same frequency as the booster's transmitter. Additionally, it had been decided that the mission would utilize the same transmitter frequency used by the earlier Landsat satellites in order that existing International Ground Stations could easily access Landsat-6. The booster telemetry data was deemed as the highest priority by all signatories to the Interface Control Document.

The Landsat-6 spacecraft was comprised of four main segments: the Instrument Mounting Platform (IPA), the Equipment Support Module (ESM), the Lower Equipment Module (LEM), and the Solar Array (SA), alternatively referred to as the Four-Pack because of its four panel design. The Payload Support Adapter (PSA) provided the structural interface between the spacecraft, at the separation plane, and the booster adapter ring. The spacecraft is depicted in Figure 2 and an exploded view of the LEM is shown in Figure 3.

Viewed as a system, the spacecraft also comprised a traditionally defined complement of subsystems: Attitude Determination and Control (ACS), Command and Data Handling (CDH), Payload (PLD), Power (PWR), Reaction Control (RCS), S-band Communications (COM), Structure (STR), and Thermal Control (THM). Figure 4 shows the spacecraft Ascent Phase block diagram.

The IPA provided a thermally stable and vibrationally isolated mounting location for the Enhanced Thematic Mapper (ETM) and the ACS instruments: the Celestial Sensor (CSA); the Inertial Measurement Unit (IMU); and the Earth Sensor Heads (ESH). Effectively, the Attitude Determination and Control subsystem's job was to point this platform such that the ETM center of scan was along the nadir at all times.

The ESM housed, or supported externally, most of the Command and Data Handling components, most of the Payload Subsystem components, and the ACS components that did not require mounting to the IPA.

The LEM housed, or supported externally, the RCS, the Power Subsystem components, the 85

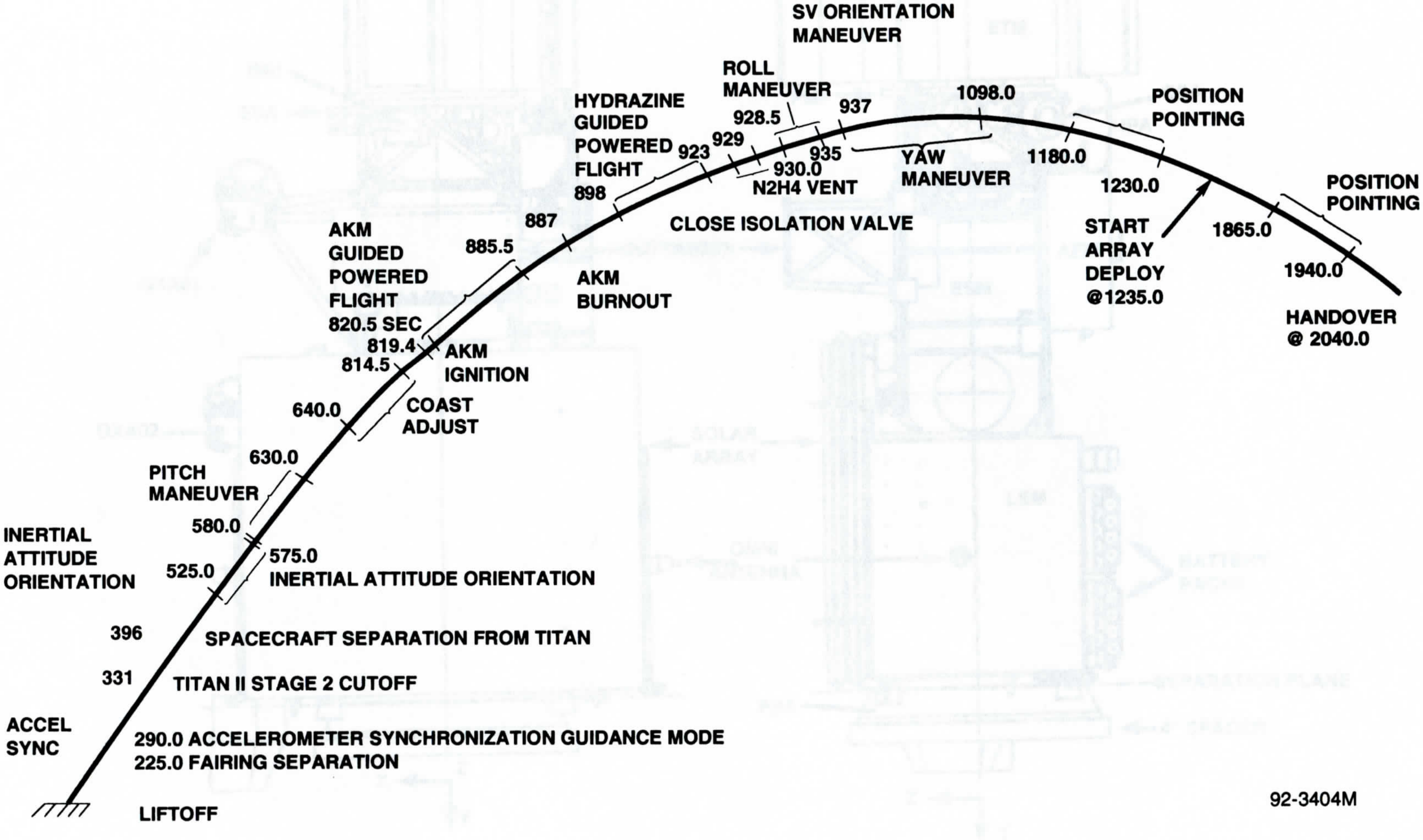


Figure 1. Landsat-6 Nominal Ascent Phase

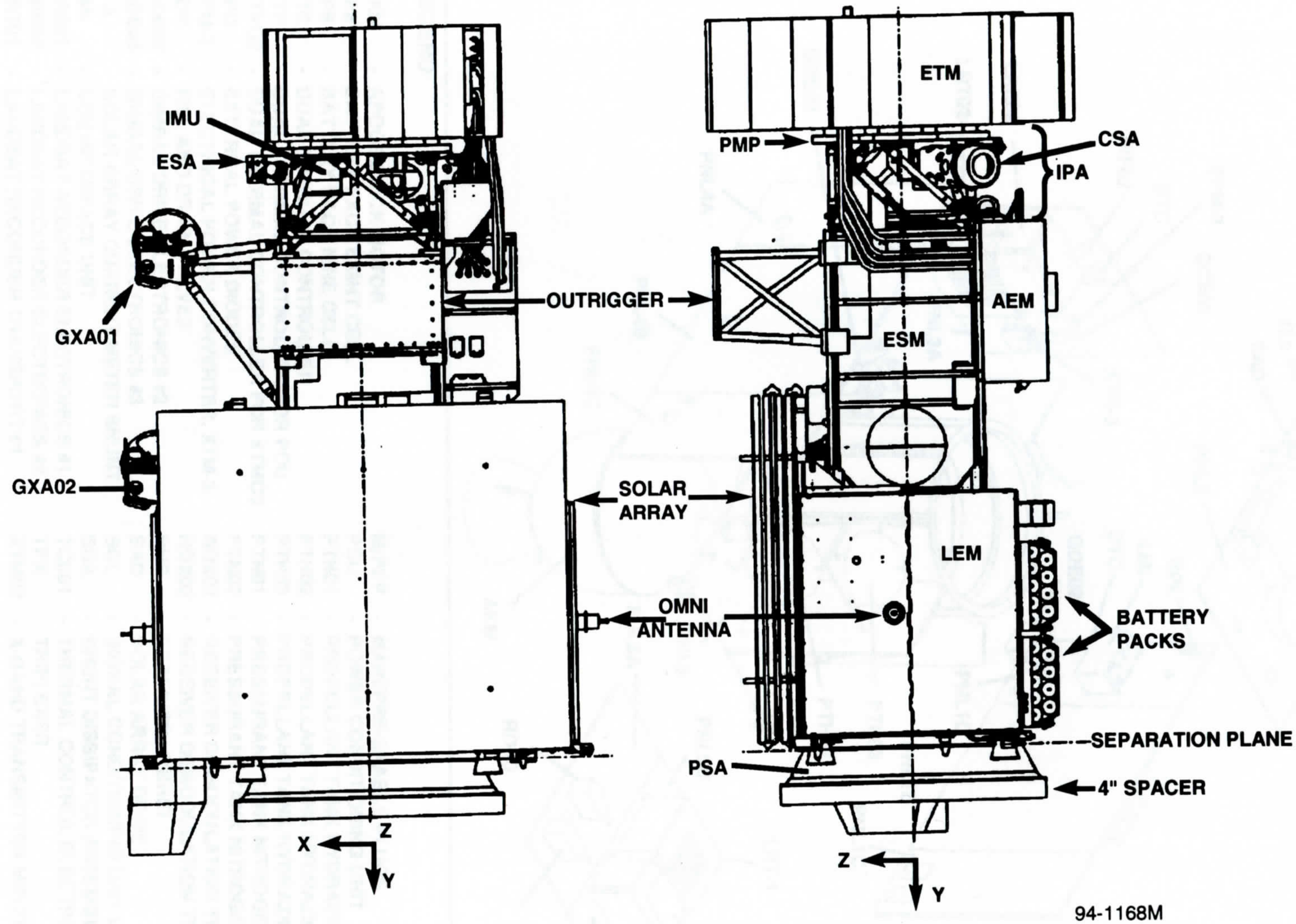
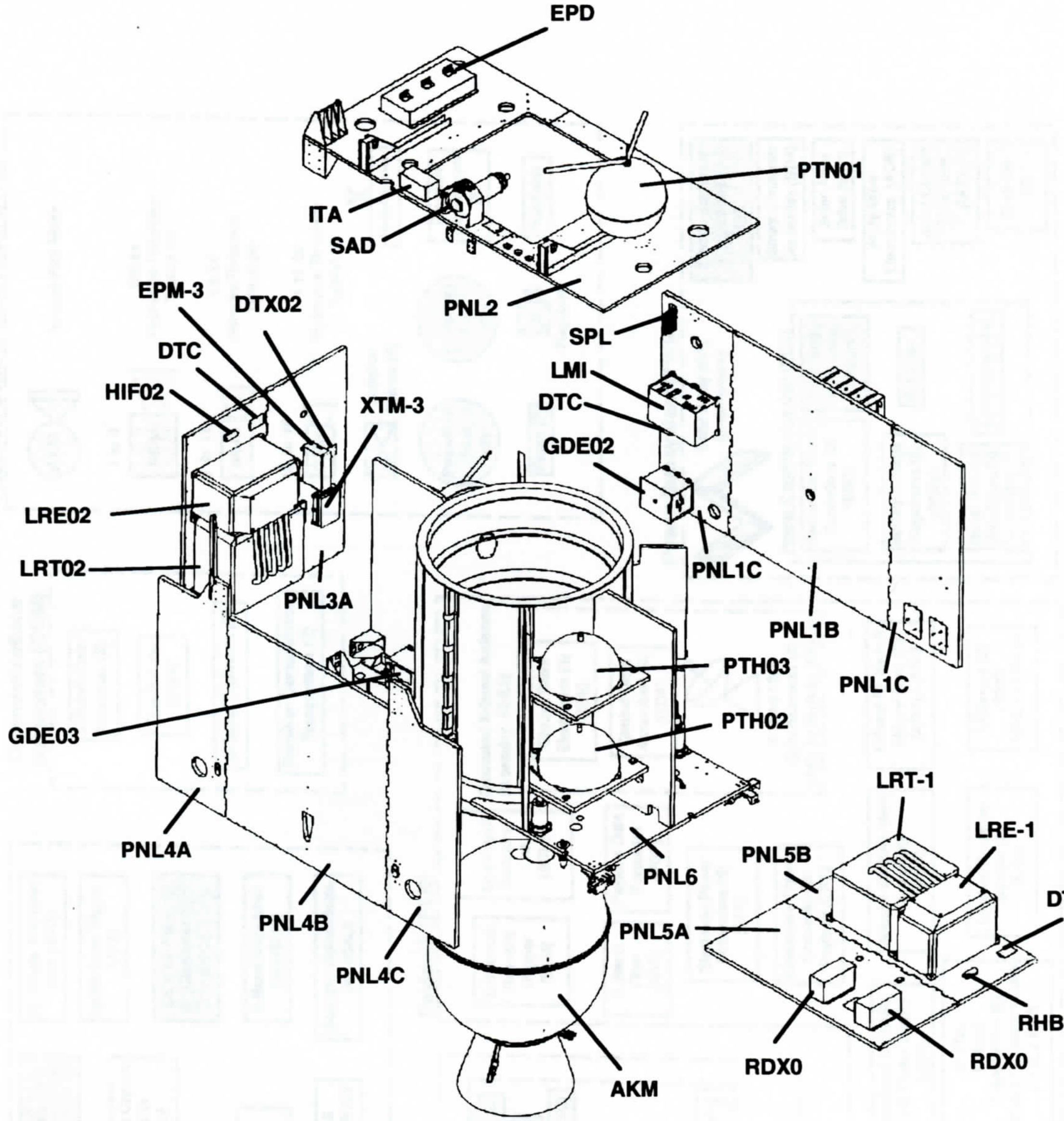


Figure 2. Spacecraft, Stowed Configuration



LEGEND

AKM	- APOGEE KICK MOTOR	M/PEP	- MAIN/PRE-ENABLE PLUGS
BP8	- BATTERY PACK EIGHT CELL	PCU	- POWER CONDITIONING UNIT
BP9	- BATTERY PACK NINE CELL	PTH01	- PROPELLANT TANK HYDRAZINE #1
DTC	- DUAL THERMAL CONTROLLER	PTH02	- PROPELLANT TANK HYDRAZINE #2
DTP	- DUAL THERMAL CONTROLLER FOR PCU	PTH03	- PROPELLANT TANK HYDRAZINE #3
DTXO2	- DUAL THERMAL CONTROLLER FOR XTMO3	PTN01	- PRESSURANT TANK NITROGEN #1
EPD	- EXTERNAL POWER DIODES	PTN02	- PRESSURANT TANK NITROGEN #2
EPM-3	- ELECTRICAL POWER CONVERTER, XTM-3	RDX01	- RECEIVER DEMODULATION TRANSMITTER #1
FDV	- FILL AND DRAIN VALVES	RDX02	- RECEIVER DEMODULATION TRANSMITTER #2
GDE02	- GIMBAL DRIVE ELECTRONICS #2	RHB	- RECEIVER HYBRID
GDE03	- GIMBAL DRIVE ELECTRONICS #3	SAD	- SOLAR ARRAY DRIVE
ITA	- SOLAR ARRAY CURRENT METER SHUNT	SCL	- SIGNAL CONDITIONING UNIT LEM
LMI	- LEM INTERFACE UNIT	SDA	- SHUNT DISSIPATOR ASSEMBLY
LRE01	- LANDSAT RECORDER ELECTRONICS #1	TCL01	- THERMAL CONTROL ELECTRONICS FOR LRS #1
LRE02	- LANDSAT RECORDER ELECTRONICS #2	TPX	- TRIPLEXER
LRT01	- LANDSAT RECORDER TRANSPORT #1	XTM03	- X-BAND TRANSMITTER MID-FREQUENCY #3
LRE02	- LANDSAT RECORDER TRANSPORT #2		

94-1170M

Figure 3. LEM Internal Configuration, -X +Z View

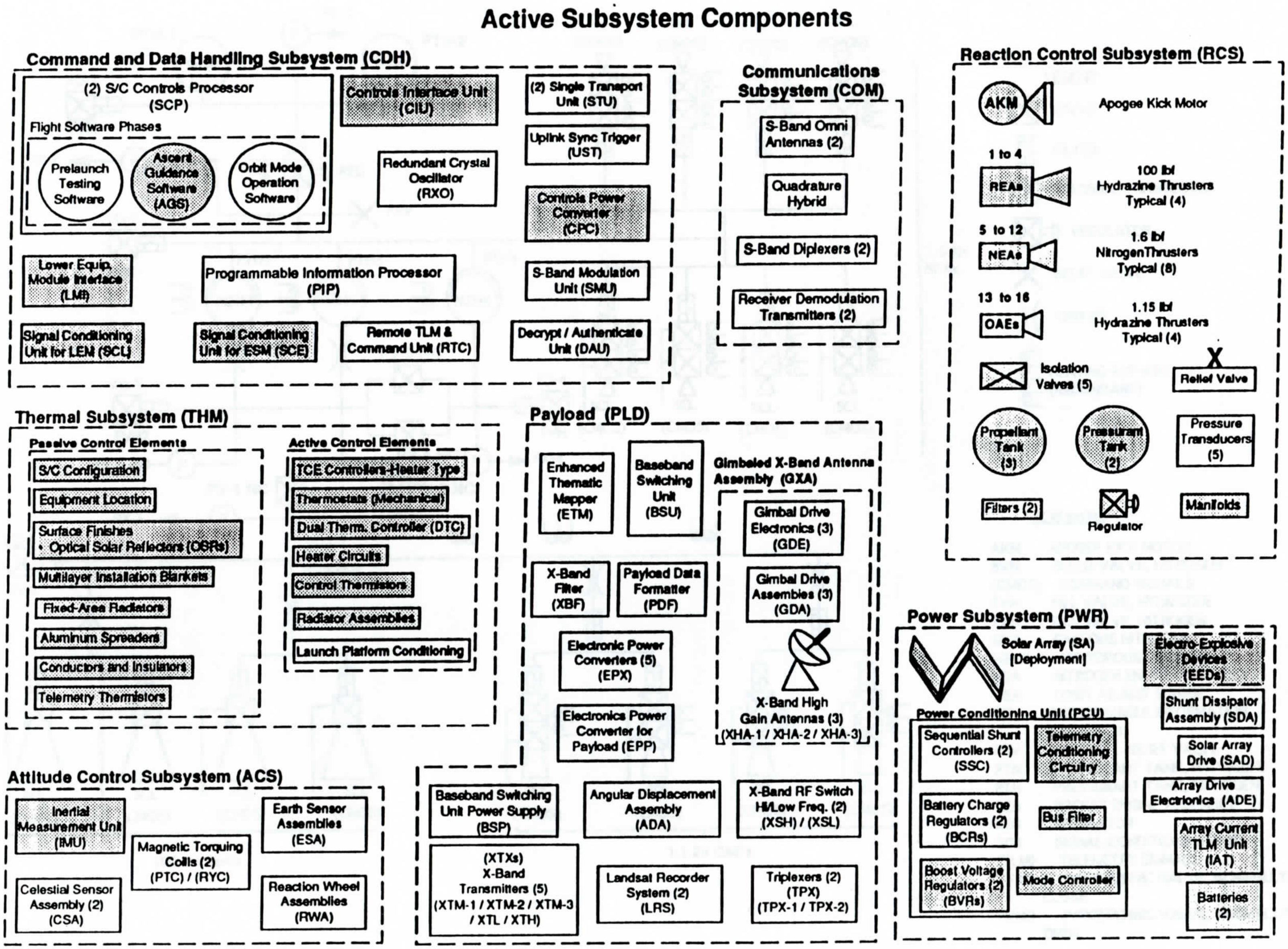


Figure 4. Spacecraft Ascent Phase Equipment Block Diagram

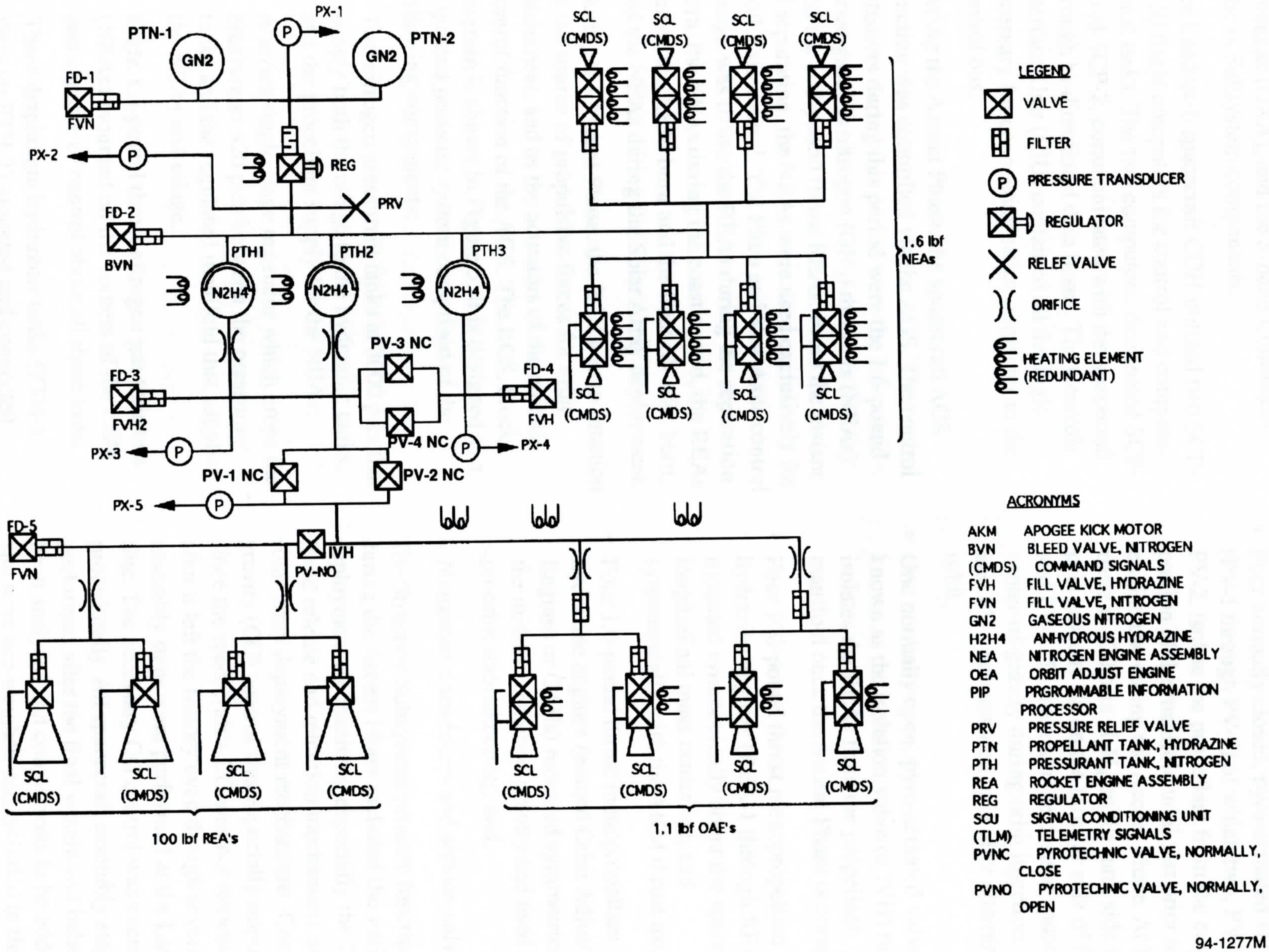


Figure 5. RCS System Block Diagram

Mbps image data recorders (Landsat Recorder Systems or LRSs), two of the Gimbaled X-band Antennas (GXA), and the S-band Communications Subsystem components.

The Landsat-6 spacecraft CDH utilized two SCP-1750 flight computers for control and computational tasks. The two computers, designated SCP-1 and SCP-2, communicated with the spacecraft through bi-directional data buses. The Controls Interface Unit (CIU) contained all the logic necessary for connecting either computer to the control bus.

During the Ascent Phase, the spacecraft ACS function was controlled via the AGS. The control actuators during this period were the 1.6-pound thrust gaseous nitrogen (GN_2) thrusters (NEAs) and the 100-pound thrust REAs. From the instant of separation, the NEAs were used exclusively for Roll Axis control. The Pitch and Yaw Axes control design was to use the REAs during the separation burn, the NEAs during the coast period, the REAs during the AKM burn and the final ΔV trim burn, and the NEAs during the Solar Array deployment.

During the Ascent Phase, the RCS was to function as the source of propulsive forces for orbit attainment, and as the actuators of the attitude control function of the AGS. The RCS, block diagram is shown in Figure 5. It is designed as a regulated pressure system comprised of the following components:

- Two nitrogen pressurant tanks at 3000 psia that supply both the ullage to the hydrazine tanks, and the propellant supply for the NEAs;
- A series single-stage regulator which drops the pressure to 420 psia between the pressurant tanks and the regulated manifold that supplies the NEAs and ullage;
- Eight 1.6-pound thrust nitrogen gas thrusters (NEAs) mounted on the corners of the LEM and capable of control about all three axes;
- Three diaphragm hydrazine tanks (PTH-1 through PTH-3) mounted and connected together in a manner to maintain the proper center-of-gravity location throughout the Ascent Phase;
- Four normally closed, pyro-actuated valves (PV-1 through PV-4) of which two, PV-1 and PV-2, isolate the propellant from the engine portion of the manifold until just prior to engine use during the Ascent Phase. At liftoff, there is 420 psia N_2H_4 on the tank side and 16 psia gaseous He on the thruster side of PV-1 and PV-2. To maintain the desired spacecraft center-of-gravity during orbit injection, PV-3 and PV-4 are actuated only after attaining orbit.
- One normally open, pyro-actuated valve (also known as the isolation valve or IVH) that isolates the REAs from the propellant manifold once the Ascent Phase is completed;
- Four 100-pound thrust monopropellant hydrazine engines (REA-1 through REA-4) mounted symmetrically about the spacecraft longitudinal mass centerline, and symmetrically with the AKM thrust axis;
- Four 1.0-pound thrust monopropellant hydrazine engines (named Orbit Adjust Engines or OAEs) mounted symmetrically to the mid-life center-of-gravity and used only for on-orbit stationkeeping; and,
- Numerous transducers and service valves.

The Structure Subsystem primary functions during the Ascent Phase included the various deployment mechanisms, especially the Clamp Band release and retention mechanism, and the solar array deployment mechanisms. Center-of-gravity (CG) control was especially important since the spacecraft's CG was not remeasured after it left the factory even though several final assembly steps were performed at the Launch Site. The necessary CG control was exercised procedurally. All spacecraft assembly steps performed after the final weight and balance had been analyzed and components to be added at the site were pre-weighed and included in the analysis. Tight limits were set in the assembly procedures to help ensure that the final spacecraft weight and center-of-gravity was closely compliant with the analysis.

The design approach for the Landsat-6 Thermal Subsystem utilizes passive thermal radiators with redundant make-up heaters. The heater circuits are separately enabled via relay logic provided by the Signal Conditioning Units in the ESM and LEM (SCE and SCL, respectively). Additionally, each heater circuit is separately fused between the relay and the heater or controller. This approach minimizes the system level effects caused by a single heater element failure. The Thermal Subsystem was completely enabled at liftoff with the exception of the primary RCS line heaters. The primary RCS line heaters are not thermostatically controlled, but rather consist of a low wattage resistive element that is always powered when the circuit is enabled. The backup RCS line heater circuit is thermostatically controlled and was powered at launch.

3.0 ASCENT PHASE EVENTS

Landsat-6 was launched with its telemetry transmitters OFF because its downlink frequency was the same as the Titan II launch vehicle transmitter. It was impossible, therefore, to directly determine the spacecraft activity during the Ascent Phase. It was decided that the mission would utilize the same transmitter frequencies as had been used by the earlier Landsat satellites for telemetry, data acquisition, and tracking by multiple International Ground Stations. However, this section describes the sequence of events as the Investigative Boards were best able to determine. The sequence resulted from studying direct evidence, from inferring events from the record of Titan II telemetry, and from assuming that the spacecraft attempted to execute the planned event sequence during the periods for which there is no directly available evidence.

Liftoff of the vehicle was 17:56:29 UTC and the Titan II second stage shutdown occurred, as expected, at 18:01:58. The Titan II issued its separation discretes at 18:02:01 as verified by Titan II telemetry. At the same time, the Titan II Attitude Control System (ACS) was enabled and acted to reduce the body rates to within the allowables for separation. The post-flight analysis of the accelerometer data concluded that there was no unusual ACS activity prior to the system being inhibited at T+392 seconds.

At 18:03:02.5 UTC, AGS was scheduled to command the Electro-Explosive Devices (EED) bus to the ARMED state. Both the receipt of at least one of the booster separation discrete signals by the spacecraft, and arming of the EED bus, are evidenced by the data indicating that separation occurred at the planned time.

Simultaneous with arming the EED Bus, AGS should have started a coarse N_2 roll program and venting the 16 psia gaseous helium within the manifold to the REAs for 0.5 seconds. The roll program was designed for stabilizing large attitude rates that resulted from the venting process.

At 18:03:03 UTC, the half-second vent program was scheduled to terminate. Simultaneously, the first pyrovalve (PV-1) was to be actuated. Firing PV-1 should have begun filling and pressurizing the REA manifold up to several hundred psia. At 18:03:04, AGS was to actuate the redundant pyrovalve (PV-2). At 18:03:05, the clamp band was scheduled for release. This was the last pyrotechnic event of the separation sequence. All of the Titan II accelerometers recorded three events at precisely the time that AGS was expected to execute the pyrotechnic events. The Investigative Board accepted this data as verification that the AGS was operating properly and that the pyrovalves and clamp band were actuated at the proper time (See Figure 6).

Simultaneous with release of the clamp band, a two-second "off-to-control" mode designed to provide full and uninterrupted thrust from all four REAs was planned. This mode was to be followed by a three-second "off-to-control" mode with narrower control bands. This sequence comprised the five-second Separation Burn, and was expected to execute with 100% efficiency

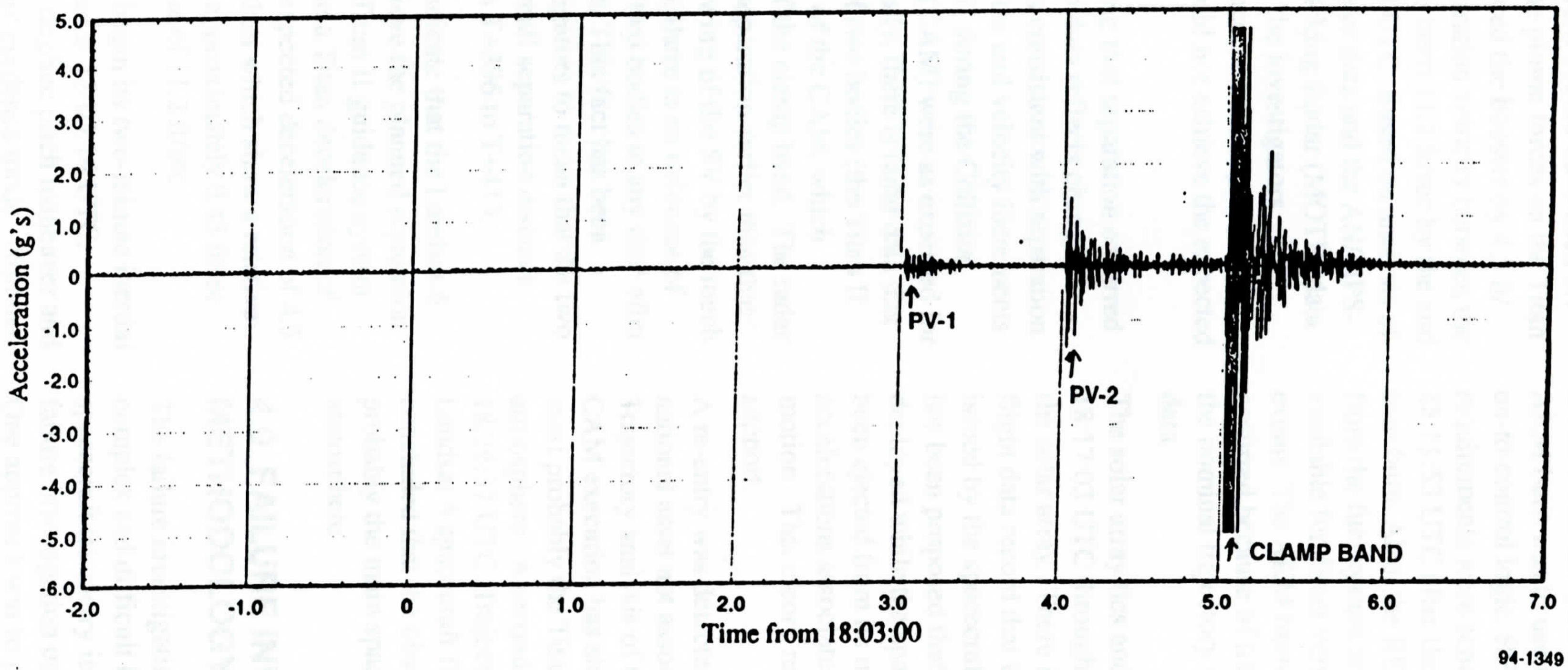


Figure 6. Titan IIG-5 2390: Accel, Tangential, Stage II SV Adapter

under normal circumstances. The analytical velocity increment imparted to the spacecraft at the conclusion of this burn should have been 6.7 ft/sec. The impact of the plume forces on the Titan II should have decelerated the booster by 4.5 ft/sec. Therefore, the separation velocity between the two bodies should have been 11.2 ft/sec by the end of the burn at 18:03:10 UTC. Based on analysis of the Titan II accelerometer data and the AN/MPS-39 Multiple Object Tracking Radar (MOTR) data for this period of time, the investigators determined that the spacecraft did separate from the booster, but that it did not achieve the expected separation velocity.

One basis for concluding that separation occurred is that the Titan II gyro data reflects changes in rotational accelerations consistent with separation. In addition, the turn rates and velocity increments achieved by the Titan II during the Collision Avoidance Maneuver (CAM) were as expected for the nominal case. Finally, there is radar data that shows a separtration of two bodies (the Titan II and SV) after the start of the CAM, which confirms the release of the clamp band. The radar could not discern the separation earlier than that probably due to shadowing of the SV by the much larger booster. Finally, there is no evidence of recontact between the two bodies at any time after the clamp band release. This fact has been accepted by the investigators to mean that the two bodies maintained a small separation distance during the period from T+396 to T+413.

Two sources of data indicate that the Landsat-6 spacecraft did not achieve the planned separation velocity. Namely, the Titan II guidance system accelerometers indicate a Titan deceleration of 0.04 ft/sec versus the expected deceleration of 4.5 ft/sec and the MOTR data which show a relative separation velocity of approximately 0.45 ft/sec versus an expected value of 11.2 ft/sec.

The spacecraft was to begin its two-minute inertial pointing guidance mode at 18:05:14 UTC, followed by a -1.8226 deg/sec pitch maneuver and two-minute coast adjust guidance mode. Nominal AKM ignition was predicted for 18:10:07 (primary) and 18:10:08 (backup) with the powered flight calculated to end at 18:11:15 UTC. Pitch and yaw axis control during the AKM burn was to use the REAs actuated through on-to-control logic. REA performance requirements were scheduled to terminate at 18:11:53 UTC after the final spacecraft velocity trim burn. Also, the REAs were to be isolated from the fuel system at that time. There is no data available for direct verification of any of these events. The AKM burn is accepted as having occurred because of trajectory analysis linking the nominal trajectory with reported re-entry data.

The solar array ties and snubbers were to fire at 18:17:03 UTC through 18:17:10 UTC and deploy the solar array. There is no positive proof in the flight data record that these commands were issued by the spacecraft computer. However, it has been proposed that the solar array could have deployed while the spacecraft was tumbling, and been ejected from its mounting because of high accelerations associated with the rotational motion. This theory remains without analytical support.

A re-entry was detected at 18:25:57 UTC by a national asset not associated with the launch. Trajectory analysis of the Titan II flight after CAM execution has shown that this event was most probably the Titan II re-entering the atmosphere. A second re-entry was observed at 18:26:37 UTC. Trajectory analysis of the Landsat-6 spacecraft flight after separation, has concluded that the observed event was most probably the main spacecraft body re-entering the atmosphere.

4.0 FAILURE INVESTIGATION METHODOLOGY

The failure investigation was extraordinarily complex and difficult because of the lack of any spacecraft telemetry information. The initial failure investigation used two parallel approaches. One approach was to perform analyses that answered specific questions from the

Investigation Boards that might narrow and focus the investigation by attempting to eliminate postulated failure modes from consideration. These activities concentrated on the RCS, the AGS, the available radar data, the booster accelerometer data,trajectory analyses, simulations and ACS margins during powered flight under normal and abnormal conditions.

In order to provide assurance that the investigation would not overlook less obvious possibilities, the second approach was a more structured method known as cause-effect ("fishbone") analysis. The fishbone approach decomposes a total system into its elements (bones) by assessing the potential of each element to cause the observed failure conditions. The decomposition process continues into layered sub-elements (finer fishbones) until the possible causal factors for each possible contributing element is established. The investigative team then attempts to prove that each potential causal factor is the actual root cause of the higher level observed failure condition. The "proving" process is accomplished by analyzing the element's plausibility or possibility given the facts of the element design, fabrication, test record as well as the records of the observed failure. The parallel approach enabled the Investigation Boards to concentrate on the main areas of suspicion while ensuring that no facet of the spacecraft escaped scrutiny.

5.0 AVAILABLE DATA

Three kinds of data were available to the Investigation Boards beyond the expected design, analyses, build, integration and test logs for the Landsat-6 spacecraft:

- Flight data
  - Booster telemetry
  - Radar
  - Associated operations
  - Reentry
- Simulation data (mathematical models)
  - Uncontrolled AKM burn
  - Separation
  - Trajectory
  - Test and flight model reaction control subsystem
  - Propulsive forces from a ruptured system
  - Water hammer analyses
- Test data
  - Reaction control system water hammer tests
  - Pyrovalve pyroshock tests
  - RCS system adiabatic detonation
  - N_2H_4 explosive decomposition at PV actuation
  - N_2H_4 material compatibility tests

Each kind of data and its associated analysis is discussed at length in the Landsat-6 Program Office Final Report. This body of information provided the means to both postulate various scenarios for the Landsat-6 failure and, in turn, critically evaluate those scenarios.

A brief description of the reaction control system post-flight ground tests is given here because of its significant role in the failure investigation. The test data related to water hammer, adiabatic detonation and N_2H_4 explosive decomposition at PV actuation were obtained in a high fidelity mock-up simulation of the flight RCS at Martin Marietta's Denver facility. The system was extensively instrumented with high frequency response pressure transducers capable of accurately measuring the short duration, high magnitude pressure spikes that an analytical transient flow model had predicted would occur. A comprehensive matrix of tests was conducted using water first to measure the magnitude and location of the water hammer pressure spikes, and then using hydrazine to see if the adiabatic compression (and therefore heating) by these pressure spikes of the helium gas in the lines was sufficient to cause the hydrazine to ignite and explode (adiabatic detonation). In most of the tests very rapid acting electro-mechanical valves were used to simulate the pyrotechnic valves used in flight. This was done because of the quick turnaround time from test to test and because of the limited supply of pyrovalves. The last series of tests, using hydrazine as the fluid, employed pyrovalves. It should be stressed that the system tests were conducted following exactly the sequence employed in flight. Specifically, (referring to Fig. 5), the manifold from the tanks to the pyrovalves (PV-1 and PV-2) were filled with 420 psia water or hydrazine, while the manifold downstream of the valves to the thrusters (REAs) were filled with 16 psia gaseous helium, simulating conditions from lift-off to beginning of the venting at 18:03:02.5 UTC for the flight (referred to in Section 3.0). Each test began with the 0.5 sec. vent of the He to vacuum through the simulated REAs and upon closing the REA valves, simultaneously activate PV-1 which releases the 420 psia liquid into the now low pressure (1.7 psia) He manifold (assuming PV-1 has functioned nominally). The liquid gets to the downstream side of PV-2 quickly as all the other lines continue to rapidly fill, with short duration pressure spikes created at all the dead ends. PV-2 thus has fuel on both sides of it before it is activated to open 1 sec. after a nominal PV-1.

The results of these tests will be discussed in Section 7.0 where the candidate scenarios are described and evaluated.

6.0 SCENARIOS COMMON CHARACTERISTICS

A set of minimum criteria for any credible failure scenario was established by the Investigative Boards. The criteria took into consideration the available flight data, analytical interpretations of that data, and expert testimony about the data. The development and investigation of possible failure scenarios created numerous action items, which in turn provided more evidence toward the resolution of the various cause-effect analyses.

Proposed scenarios were judged to be credible by their ability to explain, match, or otherwise support the minimum criteria established by the Investigative Boards. Although all of the radar data was examined, the MOTR was especially useful because of its dual skin and beacon-track data. Listed in Table 1 are the criteria and the data source for each. In addition to the criteria listed in Table 1, the Investigative Boards held the general consensus that the loss of Landsat-6 was caused by a single failure mechanism.

Although each failure scenario postulates a different mechanism as the root cause of the Landsat-6 failure, there are several common points. These common characteristics derive from the minimum criteria, and a set of commonly accepted facts. Additionally, all the remaining scenarios seek to explain the spacecraft failure with only one failure mechanism. One scenario, Scenario H, was removed from consideration because it required two failure mechanisms to match all the minimum criteria. Similarly, scenarios that require intermittent failures to satisfy the minimum criteria are not considered credible, unless the failure mechanism can be randomly applied and released throughout the scenario after its first occurrence.

In all scenarios, the controlling flight computer and ascent guidance software function properly and attempt to guide the spacecraft to orbit. One basis for this theory is the existence of three shock signatures in the telemetered booster accelerometer data, and the trajectory analysis that shows that only by firing the AKM can the spacecraft reach Re-entry Point #4. The three shocks appear coincident with the times when the AGS should have initiated PV-1, PV-2, and the primary clamp band pyrotechnic devices, indicating that the computer and AGS were functioning properly up to separation. The analytical relationship between Re-entry Point #4 and the trajectory of the coupled bodies at separation implies that the computer and AGS were able to command components in the LEM after separation. This is because the best match for Re-entry Point #4 derives from adding velocity to the spacecraft from the AKM ignition point. Further, it is not possible to match the re-entry point using all the hydrazine available to the RCS. Therefore, it follows that the AKM ignited at or near the

TABLE 1. MINIMUM CRITERIA FOR PLAUSIBLE FAILURE SCENARIOS

Criteria	Source	Significance
1. Shock spectrum present at time of expected PV-2 firing much larger than the shock spectrum present at time of expected PV-1 firing	Derived from analysis of both the Titan II structural and inertial guidance system	One of the two pyrovalve firings was anomalous
2. Presence of shock data at time of expected clamp band release	IGS accelerometer data	Ordnance power and computer systems all functional at separation
3. Low deceleration of the Titan II at separation	Derived from analysis of both the Titan II structural and IGS accelerometer data	RCS system produced little or no thrust
4. Low spacecraft velocity and separation recorded by MOTR at 440 seconds	Derived from analysis of the Titan II IGS accelerometer data	Independent verification of #3
5. Spacecraft wobble observed by MOTR after 440 seconds	Derived from analysis of MOTR data	Disturbance torques on spacecraft and RCS system could not stabilize
6. No re-contact between the booster and spacecraft after separation	Derived from analysis of both the Titan II structural and IGS accelerometer data	Positive but low "thrust" produced by spacecraft
7. Nominal Titan II CAM maneuver turn rates	Derived from analysis of the Titan II IGS gyroscope data	Spacecraft separated cleanly from booster
8. Nominal Titan II CAM ΔV	Derived from analysis of the Titan II IGS accelerometer data	Booster was a considerable distance from spacecraft
9. Timing and location of re-entry	Reported by other national assets	Booster re-entered where and when expected. Re-entry Point #4 believed to be spacecraft; Re-entry time and location consistent with AKM burn on a tumbling spacecraft.
10. Must be caused by single failure mechanism	General consensus of the Investigative Board	

nominal time and that the computer and AGS were functioning after the separation.

In all scenarios, the pyrovalves are successfully initiated. The recorded telemetry and the test data from the simulations and tests conducted on the second Landsat structure provide overwhelming evidence to support this assumption.

In all scenarios, the spacecraft and booster separate at the proper time. The booster IGS-sensed rotational rate recorded before and after the separation time supports the theory that the two vehicles separated coincidentally at the clamp band initiation. Additionally, the booster turn rates and acceleration during the CAM matched the predicted rates quite closely, indicating that the spacecraft had separated from the booster by T+413 seconds, 17 seconds after the clamp band fired.

In all scenarios, the spacecraft and booster never re-contact each other. Analysis of the booster accelerometer data, from ACS inhibit through completion of the CAM burn, supports this belief.

In all scenarios, the AKM ignites and burns normally. This view is supported by several conclusions reached during this investigation. The trajectory analysis concluded that only by firing the AKM could enough energy be added to the spacecraft body for it to re-enter at Re-entry Point #4. The Fishbone investigation of the AKM failure pathways revealed no evidence that indicated AKM failure due to design or manufacturing defect. Finally, since there was no evidence of re-contact, no mechanism for nozzle damage was identified that could contribute to early burn-out.

Finally, all scenarios postulate that the spacecraft is not in orbit. The relationship between the reported Re-entry Point #4 and the spacecraft's trajectory at separation, and the inability of national assets to find the spacecraft in any analytically related orbit preclude the possibility that the spacecraft is still in orbit.

7.0 SCENARIOS CONSIDERED

During the investigation, 11 failure scenarios were proposed for study. Each of these scenarios describes a root cause for the failure and attempts to explain the actual flight data as consequences of the proposed root cause. The scenarios were given letter designations, A through I and Z, in the order they were proposed. By the final board meeting only four scenarios remained viable explanations of the failure. These were scenarios D, G, I, and Z. However, the final meeting also contained presentations concerning scenarios A, A′, and B. All 11 scenarios are discussed in the Martin Marietta Landsat-6 Program Office Final Report; the seven scenarios discussed in the final board meeting of March 29, 1994, were:

- Scenario A; Hydraulically Induced Manifold Rupture
- Scenario A′; Hydrazine Manifold Ruptured by System Dynamics
- Scenario B; Gaseous Nitrogen Manifold Rupture or Component Failure
- Scenario D; REAs Rendered Nonfunctional by Electrical Failure
- Scenario G; Isolation Valve Closed Prematurely
- Scenario I; Hydrazine Manifold Ruptured by Adiabatic Detonation
- Scenario Z; Hydrazine Manifold Rupture Caused by an Unpredictable Hydrazine Detonation Mechanism

Table 2 compares the seven most credible scenarios against the physical evidence developed by the Joint Investigative Board from the flight-related data. Using the recorded flight data as the baseline for comparison, all the failure scenarios that match the data best involve at least one rupture in the hydrazine system, the table also shows various conditions of each of these probable failure scenarios. The conditions that differentiate the most probable failure scenarios from the other scenarios are the different accelerometer signatures recorded at the redundant PV-1 and PV-2 initiation times, the lack

TABLE 2. MOST CREDIBLE SCENARIOS

Criteria	Scenario A	Scenario A´	Scenario B	Scenario D	Scenario G	Scenario I	Scenario Z
1. Shock spectrum present at time of expected PV-2 firing is much larger than the shock spectrum present at time of expected PV-1 firing						X	X
2. Presence of shock data at time of expected clamp band release	X	X	X	X	X	X	X
3. Low deceleration of the Titan II at separation	X	X	X		X	X	X
4. Low spacecraft velocity and separation recorded by MOTR at 440 seconds	X					X	X
5. Spacecraft wobble observed by MOTR after 440 seconds	X					X	X
6. No re-contact between the booster and spacecraft after separation	X					X	X
7. Nominal Titan II CAM burn rates	X	X	X	X	X	X	X
8. Nominal Titan II CAM ΔV	X	X	X	X	X	X	X
9. Timing and location of re-entry	X	X	X	X	X	X	X
10.Must be caused by a single failure mechanism	X	X	X	X	X	X	X

X = Supporting Data/Evidence

of recontact between the booster and spacecraft, the MOTR record of velocity and separation distance, and the MOTR record of spacecraft wobble. It is seen from Table 2 that scenarios I and Z satisfy all the criteria, whereas all other scenarios are deficient in one or more of the criteria. For that reason only scenarios I and Z will be discussed in more detail; but again, the reader is reminded that all scenarios have been examined in great detail and the rationale for vindicating each of them as causing the failure is described in the Martin Marietta Landsat-6 Program Office Final Report.

SCENARIO I: HYDRAZINE MANIFOLD RUPTURED BY ADIABATIC DETONATION (FIGURE 7)

Scenario I proposes that PV-1 actuation occurred normally and that system flow and mechanical integrity were maintained until PV-2 was activated. Then, simultaneous with PV-2 actuation, an adiabatic detonation occurred that ruptured the 1/2-inch manifold, near PV-2. There were three root causes postulated for the detonation. First, preliminary analysis predicted high pressure (>3500 psi) at the interface of the two liquid columns as they flowed together in PV-2. Second, it was proposed that the waterhammer shock produced at PX-5 traveled back into the 1/2-inch manifold and added to the shocks produced by actuating PV-2. The third proposed cause was that the first waterhammer event produced an adiabatic detonation at PX-5, and its much larger shock wave added to the shock of actuating PV-2. Both the second and third root causes relied on the shock from PX-5 to increase the local PV-2 pressure above the adiabatic threshold. The scenario goes on to propose that the clamp band firing was normal and that the spacecraft was separated from the booster by the separation connector spring forces assisted by the spring forces at the cooling line interface. The two bodies were kept from recontacting each other by the venting of the hydrazine vapor from the LEM, which was preferentially directed toward the booster by the thermal blanket and LEM panel cutout configuration. Approximately 34 seconds after the PV-2 initiation, the hydrazine was depleted from the tank, but venting continued for a short time due to the evaporation and sublimation of the residual ice and liquid trapped inside the LEM. At the proper time, the AKM was ignited, from a near nominal attitude, and the spacecraft spun up and re-entered the atmosphere.

Scenario I proposes that the failure was caused by a detonation that occurred simultaneously with the initiation of PV-2, and therefore offers an explanation for the greater shock level recorded at PV-2 initiation. Inasmuch as this postulate doesn't controvert any of the recorded shock data, the scenario and data support each other.

Scenario I is very close to explaining all of the flight data. The difficulty with this scenario is in the hydrodynamic details. During the post-flight hydrazine testing with the RCS model described in Section 5.0, the PX-5 location did not show signs of adiabatic compression induced decomposition, or detonation. There was no tube distension. There was no detonation characteristic pressure buildup, but rather, the peak pressures at that location were the result of the waterhammer effect and nothing more. Pressure data obtained from the water tests are found to be consistent when compared for timing and peak pressure surges. These similarities indicate that identical phenomena were responsible for the pressure spikes at PX-5 regardless of the fluid in use.

The scenario may violate the single mechanism requirement by relying on adiabatic detonation at PX-5 to propagate to another part of the system in order to match the flight accelerometer data. Further, the test data show that the shock wave that leaves the PX-5 location and travels throughout the system is completely dissipated by the time of PV-2 initiation. However, since the Investigative Board cannot precisely determine the exact cause of the rupture, this scenario is considered feasible and probable.

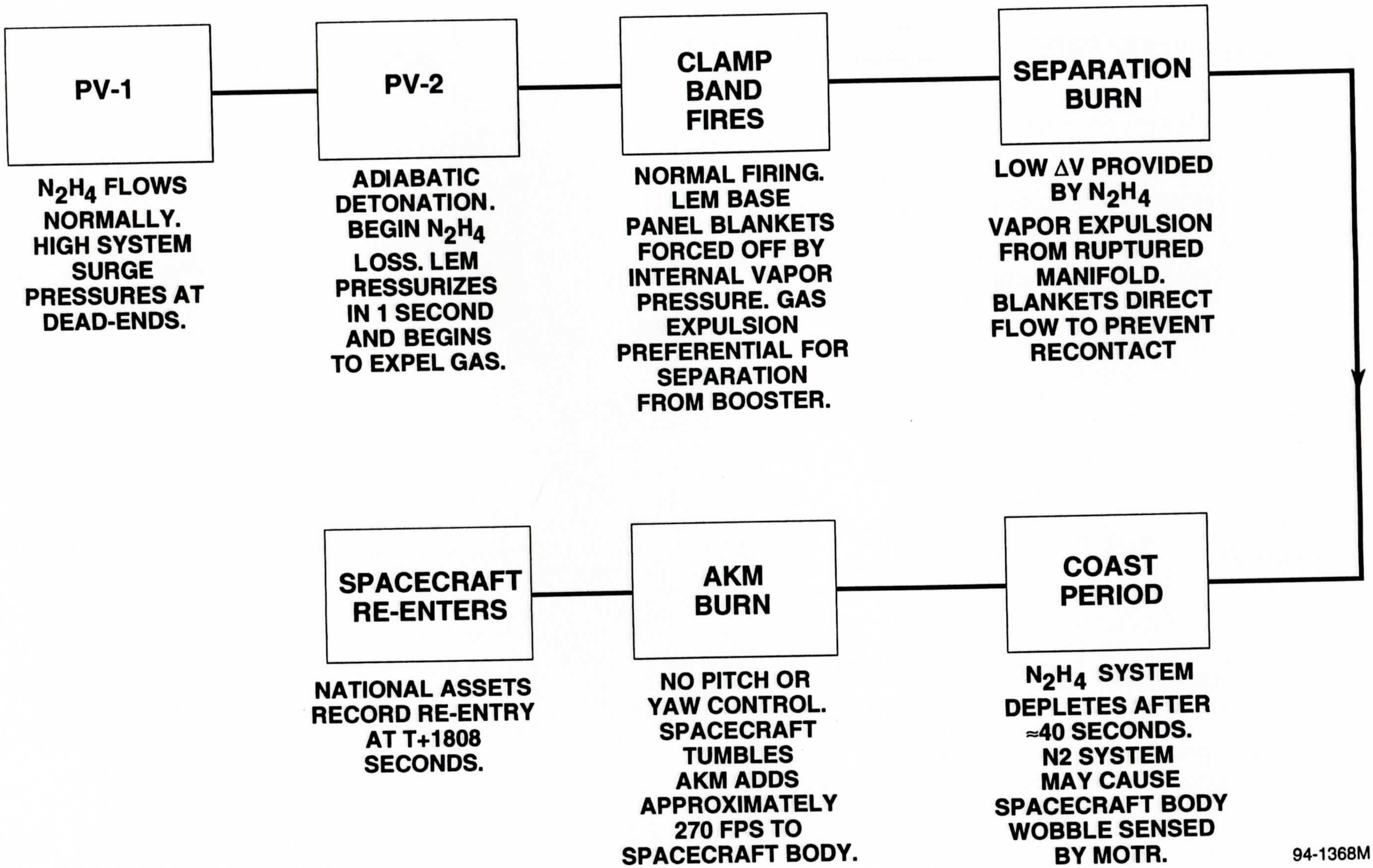


Figure 7. Scenario I, Hydrazine Manifold Ruptured by Adiabatic Detonation

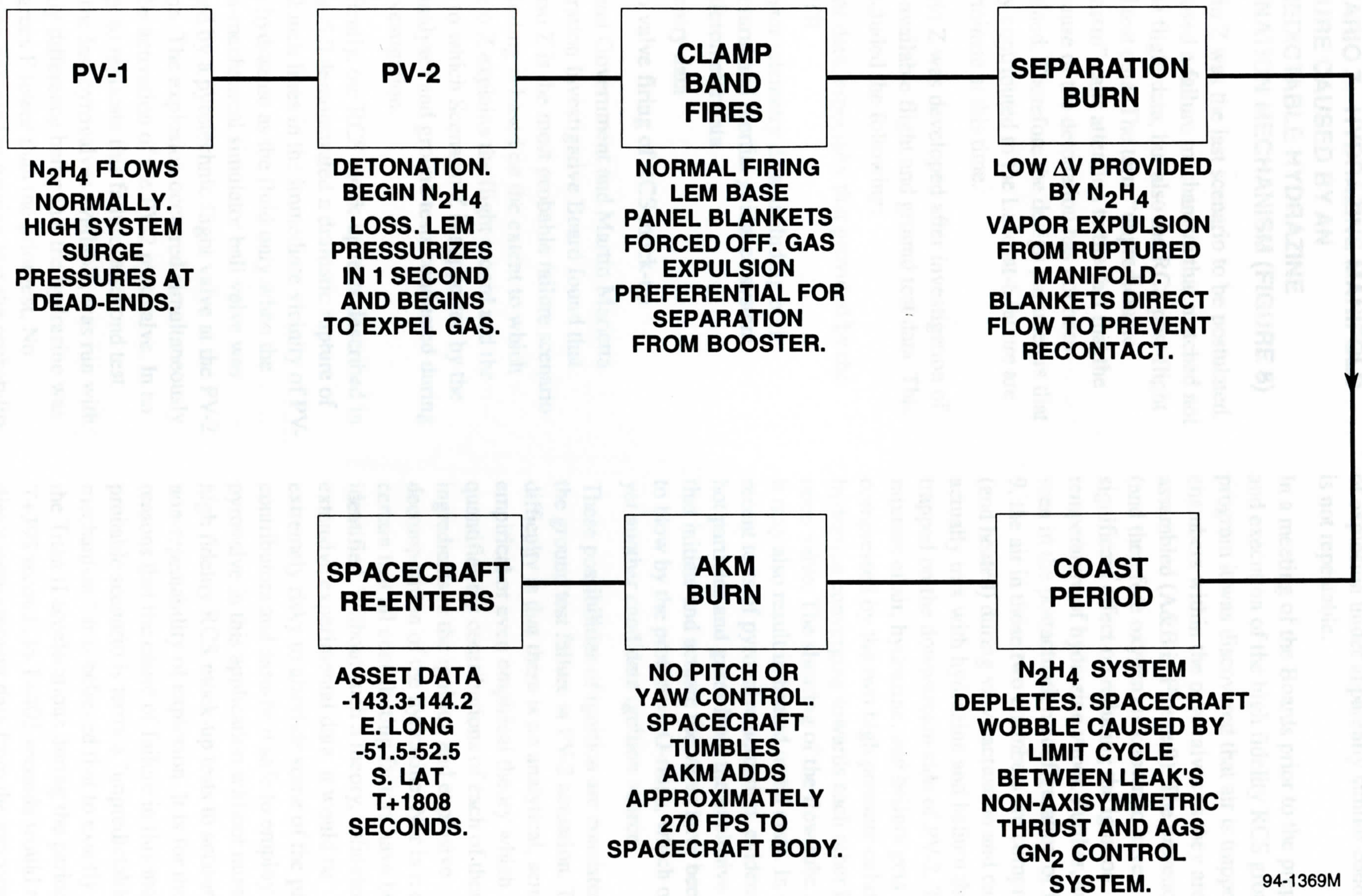


Figure 8. Scenario Z, Hydrazine Manifold Rupture Caused by an Unpredictable Hydrazine Detonation Mechanism

SCENARIO Z: HYDRAZINE MANIFOLD RUPTURE CAUSED BY AN UNPREDICTABLE HYDRAZINE DETONATION MECHANISM (FIGURE 8)

Scenario Z was the last scenario to be postulated. It proposed a failure mechanism that matched not only the flight data, but also the RCS post-flight ground test data. The term "unpredictable mechanism" calls attention to the fact that the exact cause of the detonation has not been determined. Therefore the design parameters that directly contributed to the Landsat-6 failure are indeterminate at this time.

Scenario Z was developed after investigation of all the available flight and ground test data. This data included the following:

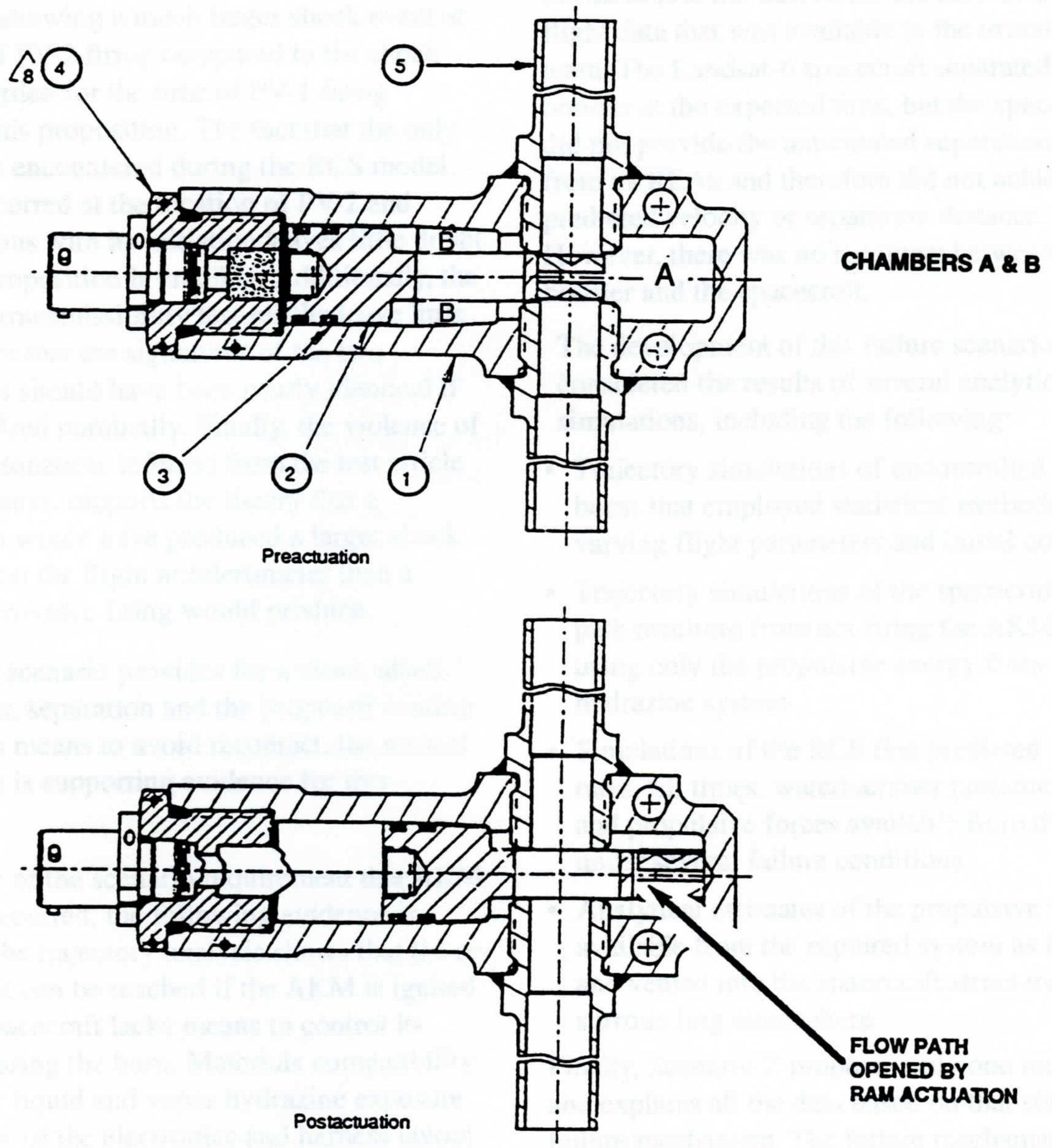
- Radar data, especially that provided by the MOTR
- Booster telemetry data, including both structural and inertial guidance system accelerometer data
- Re-entry data
- Pyro valve firing of RCS mock-up

The Joint Government and Martin Marietta Corporation Investigative Board found that Scenario Z is the most probable failure scenario. This finding is based on the extent to which Scenario Z explains the flight record and the extent to which Scenario Z is supported by the RCS analyses and ground tests conducted during this investigation.

Specifically, the RCS mock-up tests described in Section 5.0 demonstrated a dramatic rupture of the 1/2 inch lines in the immediate vicinity of PV-2 with hydrazine as the fluid only when the electro-mechanical simulator ball valve was replaced by a pyrotechnic flight valve at the PV-2 location. The explosion occurred simultaneously with the activation of the PV-2 pyrovalve. In an attempt to replicate this failure, a second test using the last pyrovalves available, was run with the only difference being that the hydrazine was 19 degrees F lower than the prior test. No explosion occurred indicating that the probability of explosion under apparently similar conditions is not repeatable.

In a meeting of the Boards prior to the preparation and execution of the high fidelity RCS ground test program it was discovered that air is trapped in chambers within the pyrovalves as they are assembled (A&B in Figure 9). The presence of air (and therefore oxygen) is known to have a significant effect on reducing the ignition temperature of hydrazine vapor. Further, as can be seen in the postactuation configuration of Figure 9, the air in those two chambers are compressed (and heated) during valve actuation and can actually mix with hydrazine and helium that was trapped on the downstream side of PV-2. This mixture of air, hydrazine, and helium gets further compressed by the two high pressure columns of hydrazine converging towards each other into the open valve. The shearing of the flow tube to open it may also result in heated metal ends. In more recent tests of pyrovalve actuation, evidence of hot particles and gases from the explosive charges that initiate and actuate the valve have been seen to blow by the protective O-rings, which offers yet another candidate ignition source.

These possibilities of ignition are consistent with the ground test failure at PV-2 actuation. The difficulty is that there is no analytical, semi-empirical or even empirical theory which quantifies the contributions of each of these ingredients to the ignition and explosive decomposition of the hydrazine. Nor is it even certain that all contributing factors have been identified. Without such a theory, validated by extensive experimental data, it would be extremely risky to alleviate some of the proposed contributors and assume it safe to employ the pyrovalve in this application without numerous high fidelity RCS mock-up tests to account for the non-repeatability of explosion. It is for these reasons that the cause of failure in this most probable scenario is termed "unpredictable mechanism." It is believed that to exactly match the Titan II accelerations during the period from T+395 seconds to T+402 seconds would require direct measurement data from the spacecraft.



List of Materials

Item	Part	Material
1	Body	15-5PH
2	Ram	13-8MO
3	O-Ring	EPR
4	Cartridge Assembly	Note 1
5	Nipple	15-5PH
Note: Booster Propellant is TiH2/KCl04		

94-1372

Figure 9. Pyrovalve Assmbly–Normally Closed

Scenario Z proposes that the failure of the Landsat-6 spacecraft was caused by a hydrazine detonation that occurred at T+395, simultaneous with the initiation of PV-2. The flight data recording showing a much larger shock event at the time of PV-2 firing compared to the shock event recorded for the time of PV-1 firing supports this proposition. The fact that the only detonation encountered during the RCS model testing occurred at the location of PV-2 and simultaneous with its initiation leaves little doubt that this proposition is credible. Additionally, the structural transmissibility test results leave little or no doubt that the signatures of the two pyrovalves should have been nearly identical if both had fired nominally. Finally, the violence of the test detonation, inferred from the test article fragmentation, supports the theory that a detonation would have produced a larger shock signature on the flight accelerometer than a normal pyrovalve firing would produce.

Since this scenario provides for a clean, albeit anomalous, separation and the proposed venting provides a means to avoid recontact, the normal CAM turn is supporting evidence for this scenario.

In support of the scenario requirement that AKM ignition occurred, the following evidence is offered. The trajectory analysis shows that the re-entry point can be reached if the AKM is ignited and the spacecraft lacks means to control its attitude during the burn. Materials compatibility testing for liquid and vapor hydrazine exposure and review of the electronics and harness layout support the conclusion that the manifold rupture would not prevent the execution of the remaining programmed events. Analysis of the ascent guidance software showed that, regardless of the spacecraft's attitude, the AKM would be fired prior to re-entry, even in the event that the spacecraft were tumbling. The spacecraft was not tumbling when last viewed by the MOTR. It is reasonable to expect the limit cycle wobble would end when all the residual liquid in the LEM had vaporized, and the venting analysis shows that all liquid is vaporized long before the nominal AKM ignition time.

Scenario Z is the best fit for the recorded booster flight data that was available to the investigation team. The Landsat-6 spacecraft separated from the booster at the expected time, but the spacecraft did not provide the anticipated separation force from its REAs and therefore did not achieve the predicted velocity or separation distance. However, there was no recontact between the booster and the spacecraft.

The development of this failure scenario considered the results of several analytical simulations, including the following:

- Trajectory simulations of uncontrolled AKM burns that employed statistical methods of varying flight parameters and initial conditions
- Trajectory simulations of the spacecraft flight path resulting from not firing the AKM and using only the propulsive energy from the hydrazine system
- Simulations of the RCS that predicted flow rates, fill times, waterhammer pressure surges, and propulsive forces available from the REAs under several failure conditions
- Analytical estimates of the propulsive forces available from the ruptured system as it burst and vented into the spacecraft structure and surrounding atmosphere

Finally, Scenario Z proposes only one malfunction and explains all the data based on that single failure mechanism. The failure mechanism is supported by the high fidelity RCS mock-up post-flight test data.

8.0 CONCLUSIONS

This investigation concluded that the Landsat-6 spacecraft experienced a ruptured hydrazine manifold. The ruptured manifold rendered the spacecraft's REAs useless because fuel could not reach the engines. The failure first manifested itself as a large shock signature as sensed by the booster instrumentation package, secondly as a low separation velocity, and finally as an inability to maintain attitude control during the AKM burn. As a consequence of tumbling during the AKM burn, the spacecraft did not accumulate sufficient energy to attain orbit and instead reentered the atmosphere south of the equator, at roughly 1808 seconds after lift-off. This conclusion is validated by the lack of Landsat signal acquisition at Kiruna, Sweden, and the observations of other national assets.

The conditions tested during this failure investigation were shown to be capable of producing an explosive event of sufficient severity to rupture the pyrovalve manifold. An 8:1 relative difference between the shocks measured by booster accelerometers at PV-1 actuation and PV-2 actuation have not been adequately explained by valve-to-valve variability or differences in the mounting of the pyrovalves to the spacecraft structure. The force of an explosion at PV-2 actuation could account for the difference.

A rupture of the 1/2-inch fuel lines at the PV-2 location was shown by analysis to be capable of reducing the fuel pressure at the REAs to virtually zero. The pressure transducers at the REA locations confirmed that there was no residual pressure in the manifold downstream of the rupture immediately after the RCS mock-up Test #11 detonation. The loss of fuel pressure prior to commanding the REAs to perform the 5-second separation burn would account for the absence of substantial separation velocity as measured by booster accelerometers and ground tracking assets.

It is probable that the exact conditions of the fluid flowing around bends and past tees influences the amount of hydrazine frothing and the relative position of the compressed helium bubble with respect to PV-2. These nonrepeatable processes could account for the lack of an explosion during Test 13. It should also be noted that the fuel temperature for Test 11 (an explosion occurred) was 71 degrees F and the fuel temperature for Test 13 (no explosion) was 52 degrees F. It is possible that the difference in fuel temperatures contributed to the variability of the results.

The investigators concluded that conditions existed that could have resulted in an explosive event at PV-2 actuation. It is not unreasonable to expect that such an explosive event occurred at PV-2 actuation during the Landsat-6 flight. The explosion would account for the high shock signature measured by the booster accelerometers and the lack of separation velocity. The resulting inability to provide control authority during AKM burn would explain the failure of the Landsat-6 spacecraft to achieve orbit.

It is beyond the scope of this effort to investigate the mechanics of how the explosion occurred or which parameters are critical to preventing such explosions. It is reasonable to conclude that the Landsat-6 failure was due to an explosive event in the hydrazine system caused by conditions not previously reported as to be capable of triggering adiabatic compression induced detonation of hydrazine.

9.0 RECOMMENDATIONS

The results of this investigation should not be limited to a review of the cause of the Landsat-6 mission failure. The findings of the Joint Investigative Board must be applied to future projects with maximum emphasis on the lessons learned from this loss. With this intention, the following sections discuss suggestions for improvements in testing and modeling a hydrazine fuel system and offer possible approaches for dissemination of "lessons learned" information.

Testing

Any newly designed hydrazine fuel feed system should be tested extensively. The test model should incorporate the actual flight sequences and flight equipment built to the flight drawings. This methodology may be the only way to mitigate the risk inherently incurred by the variability of the detonation controlling parameters. The test program should include more than one test using the planned flight sequence, flight or flight-type components, and the planned fuel at the expected environmental extremes. Particular attention should be given to the qualification and application of normally closed, pyro-actuated valves.

Models and Research

During the system design phase, it is necessary to create models that closely resemble the flight system. The design team should perform sensitivity analysis to various parameters and predict test results. They must verify the flow regimes in each line and check for cavitation and waterhammer pressures above 100 psi. The system must be designed so that gas-liquid phase interfaces are not trapped near any pyro valves when they are actuated.

A task force should be formed to address the best methodology for determining the parameters that designers must control in order to provide safe and failure-free hydrazine feed systems. The task force should enlist membership from government, industry technical staff, and academia. One likely sponsor of such a task force could be the AIAA. Once the task force issues its recommendations for the research tasks, a funding profile should be established among the corporations and government agencies that will benefit from the results. All test results should be openly shared within the aerospace community. The goal for completion of the research should be no later than December 1995, and a designers handbook should be issued no later than June 1996.

Launch

Although not related to the root cause of the Landsat-6 failure, implementation of the following recommendation would aid in the investigation of future launch or mission anomalies regardless of cause. Neither the booster vehicle nor satellite vehicle should be launched without telemetry active from liftoff to mission completion. Appropriate ground or aircraft telemetry receivers should be deployed to receive all critical events.

10.0 APPENDIX

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